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IUS SOLID ROCKET MOTOR CONTAMINATION PREDICTION METHODS

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ABSTRACT

The Inertial Upper Stage (IUS) will provide transportation from low earth orbit to synchronous orbit or for interplanetary missions for many future spacecraft. The IUS Solid Rocket Motors represent a potential spacecraft contamination source which must be considered by spacecraft designers. A series of computer codes have been developed to predict solid rocket motor produced contamination to spacecraft sensitive surfaces. Subscale and flight test data have confirmed some of the analytical results. Application of the analysis tools to a typical spacecraft has provided early identification of potential spacecraft contamination problems and provided insight into their solution; e.g., flight plan modifications, plume or outgassing shields and/or contamination covers.

INTRODUCTION

A number of spacecraft with solid rocket motors (SRM's) have experienced degradation of thermal control with a resultant shortening of operational life, apparently due to contamination. Others have flown with calorimeters, reflectance gauges or quartz crystal microbalances and have measured various levels of contamination. The amount of instrumentation and locations have been limited on all spacecraft measurements, making it impossible to differentiate between sources of the apparent contamination.

In 1978, the Boeing Aerospace Company proceeded to develop analytical tools to predict contamination associated with the solid rocket motors of the Inertial Upper Stage (IUS). This effort resulted in a series of computer codes that defined the characteristics of exhaust flow fields, the chemical species generated and the resulting contamination.

Table I summarizes all sources of contamination experienced by a spacecraft from ground operations through the end of life (EOL) of the spacecraft and the currently-known mechanisms of transport of the contaminants from the source to the receiver. This paper discusses the SRM sources and transport mechanisms. It describes the analytical tools developed and the application of these tools to a typical spacecraft.

ANALYSIS TOOLS

A number of analytical tools have been developed to define the contamination from a solid rocket motor. The definition of the chemical species, thermodynamic and flow field properties of the motor exhaust were accomplished using existing computer codes which were modified for contamination prediction purposes. Figure 1 illustrates the analysis approach and shows the various computer codes (in parentheses) used for the different parts of the flow field. The following paragraphs describe in more detail each individual computer code or analytical approach.

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Combustion Chamber

In the combustion chamber and during the nozzle expansion, the chemical species and their thermodynamic properties were calculated using the CHEM program from the Plume Interference Prediction (PIP) computer codes (Reference 1). The chamber calculations are performed initially with the condensed species considered to obtain the proper combustion properties and to determine the percentage of condensed species in the flow. All condensed species are then removed from the list of possible products being considered. The chamber calculations and subsequent equilibrium chemistry expansion are then made with a gaseous-only composition. When the thermodynamic calculations are completed, the transport properties are calculated. A matrix of thermodynamic and transport properties are generated for various assumed levels of heat transfer between condensed species and the gas. This matrix then provides the data required in the nozzle expansion process to account for heat transfer and drag between particles and gas. The code has the option of equilibrium chemistry to the nozzle exit or chemistry can be frozen at a designated area ratio. The thermochemical data are then used as an input to PIP to define the nozzle exhaust flow field.

Nozzle Inviscid Flow Field

The nozzle inviscid flow field is calculated using the PIP computer code which uses a modified method of characteristics. The PIP computer program is designed to give detailed flow field information in the supersonic region of a reacting multiphase two-dimensional or axisymmetric flow field.

The flow of a gas/particle mixture is described by the equations for conservation of mass, momentum and energy. The code is fully coupled in that it considers the exchange of momentum and energy between the gas and particle phases. The equilibrium thermochemistry data were generated as described in the previous paragraph. In the gaseous phase the state variables pressure, density and temperature are related by the equation of state while for the particulate phase the equations are for the particle drag, particle heat balance and the particle equation of state.

A particle size distribution is input in the combustion chamber and the interactions of the particles with the gas (drag and heat transfer) define a radial mass distribution and limiting streamlines for the various size particles.

From the PIP code the nozzle exit plane inviscid thermodynamic and flow field properties are defined. The exit conditions must now be modified to include boundary layer effects.

Nozzle Boundary Layer

Earlier studies (Reference 2, 3, 4 and 5) have shown that nozzle boundary layers allow the flow field to expand to much greater angles than an inviscid solution. The Boundary Layer Integral Matrix Procedure (BLIMP) computer code computes nonsimilar chemically-reacting laminar or turbulent boundary layers for ablating, transpiration cooled or non-ablating internal flows (Reference 6). The program considers either local thermodynamic equilibrium or frozen composition for a general propellant gas. The code calculates both the subsonic and supersonic parts of the boundary layer and defines the velocity and

temperature profiles. It uses the axial pressure and velocity gradients along the nozzle wall as defined by the inviscid flow field calculation. Wall temperatures from full scale development motor tests are used as input to include heat transfer at the wall.

After calculating the boundary layer the PIP code exit plane conditions are modified near the nozzle wall to account for the boundary layer thickness and velocity profile.

Nozzle Exhaust Flow Field (Past Exit)

The exit plane conditions are modified to include boundary layer effects by superimposing the profiles calculated in BLIMP. The subsonic layer is simulated using the method of Cooper (Reference 5) where the subsonic portion is isentropically accelerated to conditions slightly above sonic conditions. With the nozzle exit properties modified to include viscous effects, the PIP code is restarted to calculate the exhaust plume.

A free molecular flow calculation has been provided as an option which permits treatment of the rarified regions of the plume. As the gas expands, the translational, vibrational and rotational modes freeze based on a Knudsen number of 10, 1.0 and 0.1 respectively. At freezing, the solution switches to an effective source solution. The streamlines are considered straight and the velocity constant. Conservation of mass then determines the density while other properties are found from the equation of state.

Molecular Flow Field

The CONSIM (Contamination Simulation) computer code was developed to predict the flux of molecules which have sufficient thermal energy to escape the plume in the direction of a critical spacecraft surface.

Conceptually, the program is very simple. A spacecraft geometry is constructed by building up simple shapes which can be defined mathematically; i.e., cylinders, spheres etc. The exhaust plume is likewise defined as a cone, which hereafter will be referred to as the emitting surface. The position of this emitting surface was chosen so that on one side the flow can be described by a continuum model, which implies small mean free paths or high collision frequencies, while on the other side of the surface the flow is in a free molecular regime with large mean free paths and low collision frequencies. The properties of mean flow velocity, Mach number, flow angle and gamma were calculated using the PIP computer code and input to CONSIM as constants. Density variations were also calculated externally and assumed to vary as $1/r^2$ along the emitting surface.

After the geometry and properties of the emitting surface have been defined, a Monte Carlo technique is employed to calculate the flux of molecules hitting any surface of the spacecraft. The basic procedure is to look at the percentage of molecules which have sufficient thermal energy to escape from a random point on the emitting surface in a random direction. After a number of points and directions have been investigated, a solution for the flux is converged on.

The program calculates a characteristic thermal speed (U_c) of the flow defined as the bulk velocity of the flow divided by the most probable velocity.

The most probable velocity for several species is plotted in Figure 2. By assuming the flow is in local thermodynamic equilibrium at the emission point implies that the thermal velocity is given by a Maxwell-Boltzmann distribution and that all directions of the velocity are equally probable. Figure 3 is the thermal velocity distribution for several species at a temperature of 200^{o} K. From the equation in Figure 3 the shape of the curves are seen to be very dependant on this characteristic velocity.

Random thermal directions and velocities are chosen which are then vectorally added to the mean flow at the emission point. This resultant direction is then weighted by the fraction of molecules with this velocity and is then checked to see which, if any, spacecraft surfaces are hit. Tallies of the hits and misses for each surface are kept and after sufficient random directions and velocities are investigated, a converging solution for flux impinging on each surface is obtained.

Ambient Scattering

During SRM burns near low earth orbit, the ambient atmosphere is dense enough that exhaust products may be scattered by the oncoming ambient molecules and contaminate parts of the payload surface.

Exhaust products from the motor are scattered by collisions with the atmosphere which has a directed velocity V_{SC} (Figure 4) equal to the velocity of the spacecraft. Those exhaust particles in the volume element dV which are scattered into the solid angle Ω subtended at the volume element by the payload surface area A_{SC} will strike the payload and are assumed to stick. The model does not include self-scattering of the exhaust gas nor are multiple collisions with the ambient atmosphere considered. The amount of contamination in moleclues per unit area striking the payload surfaces is given by:

$$C = \int \frac{S(t,\theta)}{\rho^2} \; \frac{\nu(t,\phi)}{A_{SC}} \; \frac{\Omega(\rho,\theta)}{4\pi} \, dV\, dt \qquad (1)$$

where:

$\frac{S(t,\theta)}{\rho^2}$ is the source term which describes the density of exhaust-gas molecules as a function of time t, angle θ, and radial distance ρ.

$\frac{\nu(t,\phi)}{A_{SC}}$ is a term describing the collision frequency as a function of time and which also takes into account the angular ϕ dependence of the collision cross section of the exhaust molecules with the ambient gas. A_{SC} is the area of the payload.

$\frac{\Omega(\rho,\theta)}{4\pi}$ is a geometrical term representing the ratio of the solid angle subtended by the payload to the total solid angle 4π.

$\int dV\, dt$ represents integration over all appropriate volume and time.

Following burnout the exhaust flow field is assumed of the form:

$$\frac{S(t,\theta)}{\rho^2} = \frac{A(t)\cos\theta}{\rho^2} \qquad (2)$$

where A(t) is function of time as the internal insulation cools and outgassing slows.

The collision frequency term is given by (3)

$$\nu(t,\phi) = N(t)\,\sigma(\phi)\,V_{SC}(t)$$

N(t) is the particle density of the ambient atmosphere and is a function of time because the spacecraft is changing altitude.

V_{SC} is the spacecraft velocity and $\sigma(\phi)$ is the cross section for collisions between the exhaust gas and the ambient atmosphere, and

$$\sigma(\phi) \simeq 4 \cos\phi\ \sigma_{cm} \tag{4}$$

where σ_{cm} is the cross section measured in the center of mass frame of the colliding particles.

By determining the solid angle subtended by the volume element and substituting for the volume element, the integral can be rewritten. During burnout when the vehicle may be pointing the outgassing nozzle into the wind, the integral is:

$$C = K \int_{t_{retrograde}}^{t_{posigrade}} A(t)\, V_{SC}(t) N(t)\,dt \int_{0}^{\pi/2} \int_{\frac{R_e}{d \sin\theta}}^{\infty} \frac{(X\cos\theta+1)\cos\theta \sin^2\theta\ X}{(X^2+1+2X\cos\theta)^2}\, dX d\theta \tag{5}$$

where $K = \dfrac{2\sigma_{cm}}{d}$ and $X = \rho/d$

Similarly, during main motor burn, an equivalent integral can be constructed by modifying the source term, geometry and integration limits.

Heat Soak Outgassing

During and after burnout of a solid rocket motor, heat from the combustion chamber conducts through the insulation to the motor case materials and heats them to temperatures which can significantly increase the case material outgassing rates resulting in severe contamination to spacecraft. The analysis approach is to use a one dimensional heat conduction equation to define temperatures vs. time on the external surface of the case. Test data are required to define the volatile condensible materials (VCM) which will emanate from the motor case at the predicted operational temperatures. Table II summarizes the information required and the assumptions for this analysis.

TYPICAL SPACECRAFT CONTAMINATION ANALYSIS

The analytical tools described in the previous paragraphs were applied to the IUS and a typical spacecraft as shown in Figure 5. Typical flight operations which affect contamination are the 111 second SRM1 burn, the 5 hour 10 minute coast with the SRM1 attached, SRM1 - 2 separation, SRM2 burn of 77 seconds and the contamination/collision avoidance maneuver occuring 55 minutes after SRM2 burnout. The following paragraphs describe the results of a steady state and transient analysis of the inter-molecular scattering in the SRM exhaust plume, ambient scattering during SRM operation in low earth orbit, ambient scatter of internal insulation during a possible retrograde maneuver and outgassing of the Kevlar epoxy motor case following SRM burnout and heat soak.

Steady State

The steady state analysis was made assuming the SRM operated at a constant 500 psia chamber pressure. The exhaust flow field was defined with and without a boundary layer in the nozzle as shown in Figure 6. The boundary layer thickness is approximately 0.1 inches thick. A boundary between free molecular flow and continuum flow was defined in the exhaust flow where the Knudsen number is ten. It was assumed that a majority of the molecules which would escape the exhaust plume would eminate from this location. Using the properties of the exhaust flow field along this surface and assuming an average gas molecular weight, the deposition of contamination along the vehicle surface is shown in Figure 5. This is equivalent to a molecular layer on approximately 0.00005 percent of the surface area of the spacecraft. Since the exhaust flow field contains a large number of different molecules of various molecular weights, (Table III), and thermal molecular velocities are exponentially dependant on molecular weight, consideration was given to the effects of molecular weight on inter-molecular scattering. Hydrogen gas has the lowest molecular weight of the species and would (if the gas were to stick) put about 300 molecular layers of hydrogen on 100 percent of the surface. Assuming molecular carbon with a molecular weight of 12 is in the flow, an equivalent of one molecular layer over 0.005 percent of the surface area of the spacecraft results.

Transient Analysis

A simplified transient analysis was conducted because inter-molecular scattering increases exponentially with decreasing mean flow velocity during ignition and burnout transients. Steady flow fields were defined for transient chamber pressure conditions of 1, 30, 60, 100 and 200 psia. The ignition transient duration is approximately 0.15 seconds. The burnout transient, when considering the insulation outgassing after motor burnout, will last for four to five minutes (see Figure 7). This decay rate is defined using a combination of rubber insulation outgassing test data and analytical procedures as outlined in Reference 7. Inter-molecular scattering was defined for each of the chamber pressure conditions as shown in Figure 8 for SRM1 and 2. As the chamber pressure decreases a significant increase in molecular flux to the spacecraft is noted which peaks and begins to decline as the flow approaches a free molecular condition. SRM2 flux is small because of the smaller motor.

A summary of the ignition and burnout transients and the steady state flow field contamination is shown in Figure 9. It is evident that the transient burnout is the largest contributor to spacecraft contamination.

Ambient Scattering (Main Burn)

In the lower atmosphere (150-300 Km), the ambient environment is sufficiently dense to scatter the exhaust gas molecules in the backflow region during SRM1 operation. The ambient scattering model described earlier was used. Two atmospheric models were used, Reference 8, for the mean density atmosphere and Reference 9, for the maximum density atmosphere. Using a collision cross section of 5.0×10^{-15} cm^2, a molecular radius of two Angstroms and a typical trajectory, the ambient scattered molecules produce 1.51×10^{-5} microgram/cm^2 of contamination on the spacecraft.

Ambient Scattering (Retro-Grade Maneuver)

From a Titan Launch, the SRM1 boosts the IUS and typical spacecraft from low earth orbit toward synchronous orbit. Following SRM1 boost, a velocity vector correction may be required using the aft facing RCS pitch and yaw motors. If a reduction in velocity is required, the IUS will be rotated 180 degrees to point the RCS system to reduce spacecraft velocity. During this time, the SRM1 outgassing is decaying exponentially. The oncoming ambient molecules will impinge upon the outgassing molecules and some will be scattered toward the spacecraft. The amount of deposition of contaminant is strongly a function of both the motor outgassing rate and ambient density both of which are decaying rapidly with time. Figure 4 shows the ambient scattered contamination occuring during an SRM1 retrograde maneuver. It is evident that a delay in the retrograde maneuver can significantly reduce contamination. Currently a 180 second time delay is used for the IUS.

SRM Heat-Soak

The external surfaces of the Kevlar epoxy motor cases of the IUS SRM's are expected to reach temperatures of 500°F 25 to 30 minutes after motor burnout from heat being conducted from the inside surfaces of the motor case. Tests conducted on Kevlar epoxy materials heated to 400°F and 600°F have measured volatile condensible material (VCM) of 0.03 percent and 1.4 percent respectively. An interpolated value of 0.8 percent has been assumed for a 500°F external wall temperature. Table II summarizes the results of the analysis and the assumptions. Most of the outgassing will condense on cool internal surfaces of the interstage and equipment support ring with very little being vented overboard. The only outgassing which could reach the spacecraft would be through the interface connector bracket which is predicted to be 0.42 micrograms/cm² from the SRM2 motor case. SRM1 remains with the payload until synchronous orbit is reached which is approximately five hours after SRM1 burnout. Temperatures of the motor case at this time are predicted to be approximately 350°F which will produce only minor outgassing (VCM = 0.03%). Contamination deposition to the spacecraft from the SRM1 motor case following separation will be negligible since the temperatures have dropped significantly and the time between separation and SRM2 ignition is only three minutes.

Summary of Contamination Flux

Contamination to spacecraft results from ground operations through spacecraft end of life oprations. This report has addressed potential sources of contamination from the IUS solid rocket motors during transfer orbit operations. Table IV shows a summary of the levels of contamination from the various SRM sources. Note that the retrograde maneuver is delayed 180 seconds to reduce contamination and the contamination/collision avoidance maneuver precludes contamination from the SRM outgassing after separation by not allowing the SRM nozzle to point in the direction of the spacecraft.

ANALYSIS VERIFICATION

The amount of ground or flight data useful for verification of analytical tools is very limited and incomplete. The following paragraphs describe Lockheed and AEDC ground test data and GSFC flight data which provides some verification of the analytical tools.

LMSC Test Data

Lockheed Missiles and Space Company conducted subscale solid propellant motor tests with aluminum loadings of 2, 10 and 15 percent at simulated altitudes of 50,000, 100,000 and 112,000 feet for comparison with analytical predictions using the PIP code. The tests were conducted in the High Reynolds Number Wind Tunnel Test Facility at the Marshall Space Flight Center. The details of the test and analyses are found in Reference 10 and an example of the comparison is shown in Figure 10. This figure shows that the computer code predicted a drop in the pitot pressure radial distribution with an increase in aluminum concentration which correlated with a measured drop in pitot pressure from the test program. It would indicate that the computer code is correctly accounting for the gas-particle interactions on total pressure loss.

AEDC Bi-Pr pellant Data

In March 1978, AEDC reported data (Reference 11) taken on a 5-lbf bi-propellant motor for gas flux measurements in the plume backflow. The motor used mono-methyl hydrazine (MMH) and nitrogen tetroxide (N_2O_4) as the fuel and oxidizer, respectively. Many combinations of oxidizer/fuel ratio, nozzle expansion ratio, chamber pressure, chamber geometry and duty cycle were investigated to determine their effects on the flux. A standard configuration was chosen which had oxidizer/fuel = 1.6, expansion ratio - 100, chamber pressure - 100 psia, 2-inch cylindrical combustion chamber and one percent duty cycle of 100 msec pulses. The baseline configuration was placed in the AEDC 10 ft. diameter by 20 ft. long cryogenic test chamber and the backflow fluxes were measured by eight quartz crystal microbalances (QCM's). The QCM's were cooled to 25^{o}K and were placed 26 to 147 degrees from the thrust axis and 39.4 to 155 cm. from the nozzle exit. Figure 11 shows the test data plotted against the flux calculated using the gasdynamic codes described previously. Reasonable correlation is obtained between the analysis and test data to approximately 135 degrees.

GSFC Flight Data

The Anchore Interplanetary Monitoring Platform (AIMP-E) was launched in 1967 with a contamination monitor on the fourth stage near the THIOKOL TE-M-458 retro motor (Reference 12). The contamination monitor consisted of a light source, a reflectance plate and a solar cell sensor. The configuration is shown in Figure 12 along with the measurements made by the instrument. Approximately three minutes after fourth stage burnout, the contamination monitor indicated a change in the absorptivity which continued for 15 minutes. The absorptivity of the reflecting surface changed from 0.1 to 0.25 during this time. The decreasing absorptance at 9 hours and 30 minutes is not understood. The total deposit is known to have remained approximately two hours before the data recording was terminated.

A number of observations should be made: 1) no contamination was noted during solid rocket motor firing (25 seconds) which would indicate that the contamination was not associated with intermolecular self scattering during motor firing, 2) the motor case started to increase in temperature at about 3 minutes after burnout and peaked at 21 minutes after motor burnout, indicating the apparent contamination could be from materials outgassing outside the motor case (mylar insulation covered motor area) , and 3) the contamination accumulation 3 to 18 minutes after motor burnout is consistent with the transient analysis which predicts the majority of contamination from intermolecular interactions occurs after motor burnout.

CONCLUSIONS

1. Analytical tools have been developed to predict spacecraft contamination from solid rocket motors. Limited ground and flight data have confirmed some of the analysis tool results. LMSC subscale test pitot pressure data shows a reduced total pressure with increasing aluminum particle concentration which is confirmed by the analytical tools. Analytical predictions of mass flux in the plume back flow region correlate with AEDC experimental measurements. AIMP-E flight data indicates significant contamination on a contamination monitor after motor burnout which correlates with the analytical predictions.

2. Application of the analysis tools to a typical spacecraft on the IUS have produced the following conclusions:

   a. The outer surface of the IUS Kevlar epoxy motor case reaches 500°F 25 to 30 minutes after motor burnout which allows 0.42 micrograms per cm^2 direct flux to the spacecraft through the interface electrical connector plate. The thermal blanket at the IUS/spacecraft interface prevents significantly more contamination to the spacecraft.

   b. A delay in retrograde maneuver of 180 seconds following SRM1 burn reduces the spacecraft contamination from 2.6 microgram/cm^2 to 0.015 micrograms/cm^2. This contamination results from ambient molecules scattering SRM1 outgassing products toward the spacecraft during a negative velocity correction maneuver. No delay is required in the retro maneuver for SRM2 since the ambient density in synchronous orbit produces negligible scattering.

   c. Intermolecular (self) scattering of exhaust products toward the spacecraft during or following SRM burns is negligible. The burnout outgassing phase produces 1×10^{-8} micrograms/cm^2 compared to 1×10^{-18} micrograms/cm^2 during steady state burn.

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Table I: Contamination Sources

CONTAMINATION SOURCE	CONTAMINATION TRANSPORT PHENOMENA			
	DIRECT FLUX	AMBIENT SCATTERING	SELF SCATTERING	ELECTRO STATICS
o GROUND OPERATIONS				
o LAUNCH VEHICLE NON-METALLIC MATERIALS OUTGASSING				
o LAUNCH VEHICLE PROPULSION OPERATIONS AND CONTROL				
o SPACECRAFT NON-METALLIC MATERIALS OUTGASSING				
o IUS NON-METALLIC MATERIALS OUT-OUTGASSING				
o IUS SRM-1 AND -2 BURNS				
A IGNITION TRANSIENT	x	x	x	
B STEADY-STATE	x	x	x	
C BURNOUT TRANSIENT	x	x	x	
D SRM INTERNAL INSULATION OUTGASSING	x	x	x	
E SRM HEAT SOAKOUT MATERIALS OUTGASSING	x	x	x	
o IUS/RCS OPERATIONS				
o SPACECRAFT FREEFLIGHT				

x Indicates Contamination Sources Considered in this Paper

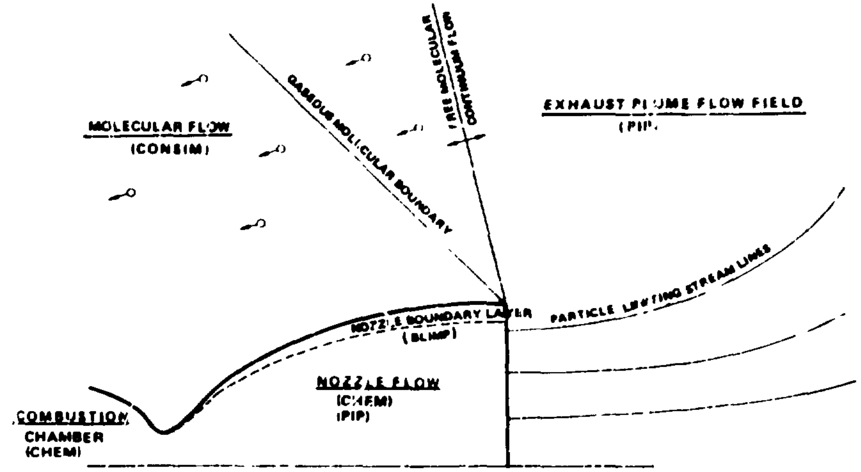

Figure 1: Analysis Approach

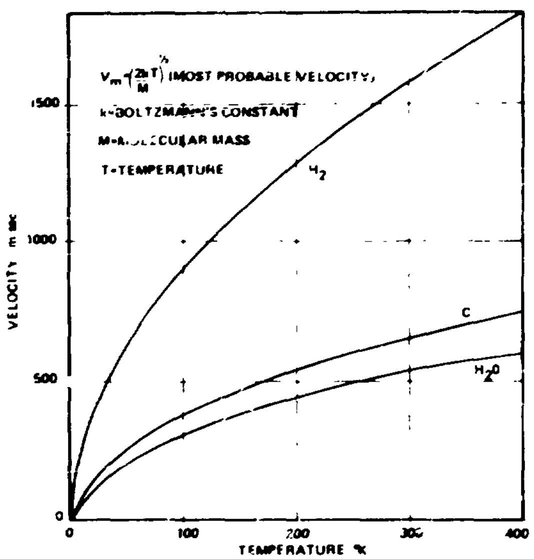

Figure 2: Most Probable Thermal Velocity Versus Temperature

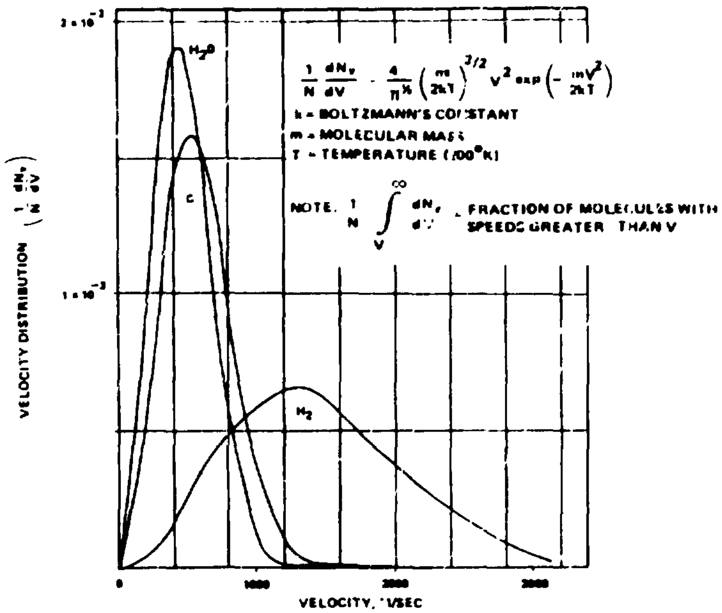

Figure 3: Thermal Velocity Distribution for Species

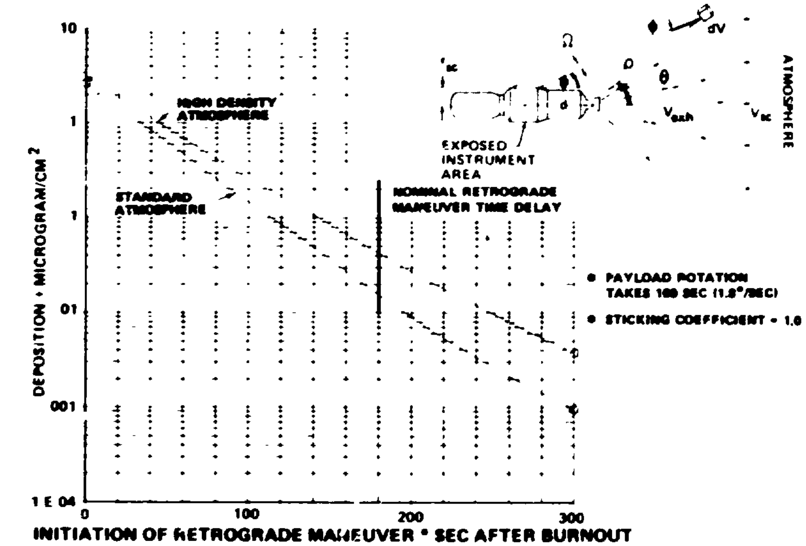

Figure 4: Contamination Deposition as a Function of Retrograde Maneuver Initiation Following SRM 1 Burnout

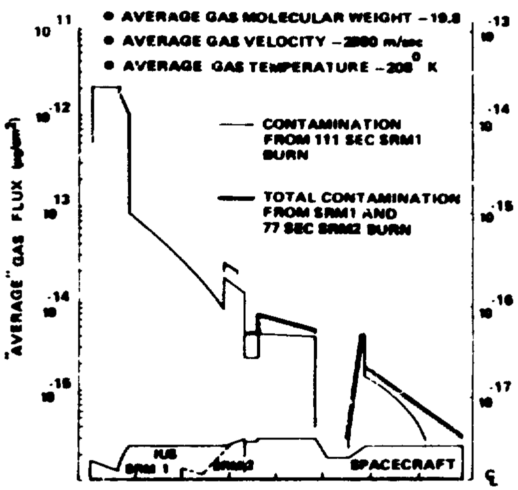

Figure 5: Average Exhaust Gas Molecular Deposition on IUS and Spacecraft

Table II: VCM from SRM1 and 2 Kevlar Epoxy through Interface Bracket

STAGE	TOTAL WEIGHT LBS (GM)	VCM IN INTERSTAGE LBS (GM)	INTERSTAGE SURFACE AREA CM^2	INTERSTAGE SURFACE AREA TO ESS VENT AREA RATIO	ESS SURFACE AREA CM^2	INTERFACE BRACKET VENT AREA CM^2	VCM TO S/C Å
*SRM1	777 2 ($3\ 53 \times 10^5$)	1.775 (805)	$2\ 75 \times 10^5$	3.6×10^6	1.74×10^5	23.87	0604 (000006 GM)
SRM2	198.2 ($8\ 99 \times 10^5$)	.6788 (308)			1.74×10^5	23 87	42 (021 GM)

*SRM1 CASE OUTGASSING CONDENSES ON INTERSTAGE AND ESS SURFACES.

ASSUMPTIONS

- 1/3 OF KEVLAR EPOXY VCM WILL OUTGAS INTO INTERSTAGE AREA FROM SRM1 HEATSOAK FOLLOWING MOTOR BURNOUT
- KEVLAR CASE OUTER SURFACES REACH 500°F TEMPERATURE
- ALL OUTGASSED MATERIAL GETS OUT OF MLI COVERING MOTOR CASE
- OUTGASSED MATERIAL WILL CONDENSE ON INTERSTAGE SURFACE (ASSUMING ITS TEMPERATURE IS APPROXIMATELY 25°C)
- AMOUNT OF MATERIAL WHICH FLOWS THROUGH VENTS IS PROPORTIONAL TO THE FRACTION OF THE VENT AREA TO THE TOTAL COMPARTMENT SURFACE AREA
- 1/2 OF KEVLAR EPOXY MATERIAL WILL OUTGAS INTO ESS COMPARTMENT FROM SRM HEATSOAK FOLLOWING MOTOR BURNOUT
- 50% OF VCM IS EMITTED DURING FIRST 1/2 HOUR 99% AFTER 4 HOURS.
- VCM CONDENSES ON 50,670 CM^2 SURFACE AREA (BASED ON S/C DIAMETER)
- DENSITY OF VCM IS 1 GM/CM^3

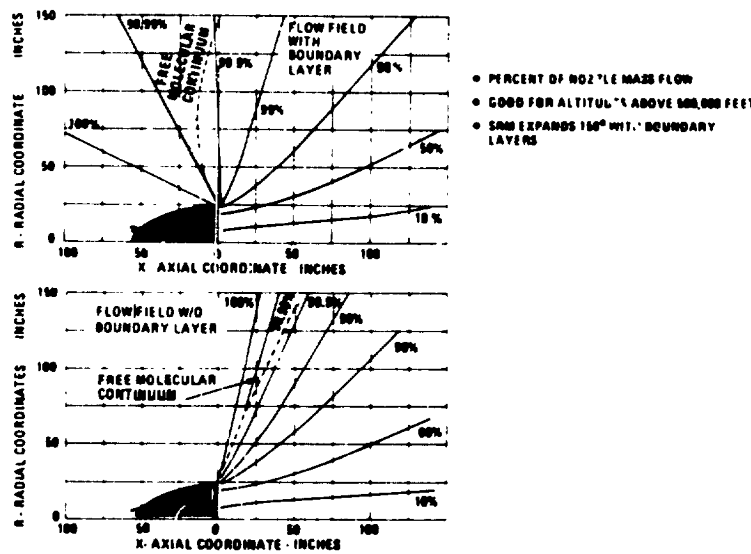

Figure 6: SRM1 Exhaust Plume with and without Nozzle Boundary Layer

Table III: Potential Species in Backflow Region

SPECIES	MOLECULAR WEIGHT	$\frac{n_x}{n_{19.8}}$ *
H_2	2	4 36 E+18
C	12	4 04 E+08
NH_2	16	2 06 E+04
NH_3	17	1 71 E+03
H_2O	18	1 41 E+02
Average Gas	19 8	1 0
Na	23	5 15 E-04
HCN	27	2 16 E-08
CO	28	1 73 E-09
N_2	28	1 73 E-09
Cl	35 5	9 82 E-18
HCl	36 5	7 77 E-19
CO_2	44	4 09 E-27
AlOH	44	4 09 E-27
Fe	56	2 13 E-40
NaCl	58 5	3 60 E-43
AlO_2H	60	7 81 E-45
AlCl	62 5	1 32 E-47
Al_2O	70	6 19 E-56
KCl	74 5	6 19 E- 61
AlOCl	78 5	3 13 E-62
FeCl	91 5	7 68 E-80
$AlCl_2$	98	4 45 E-87
Al_2O_3	102	1 57 E-91
$AlCl_3$	133.5	1 23 E-126

$$* \quad \frac{n_x}{n_{19.8}} = \left(\frac{M_x}{M_{19.8}}\right)^{3/2} e^{-(M_x - M_{19.8})\,V^2/2KT}$$

V = 2990 M/Sec (Flow Mean Velocity)
T = 208°K
M = Molecular weight
K = Boltzmann's constant
n = Number density

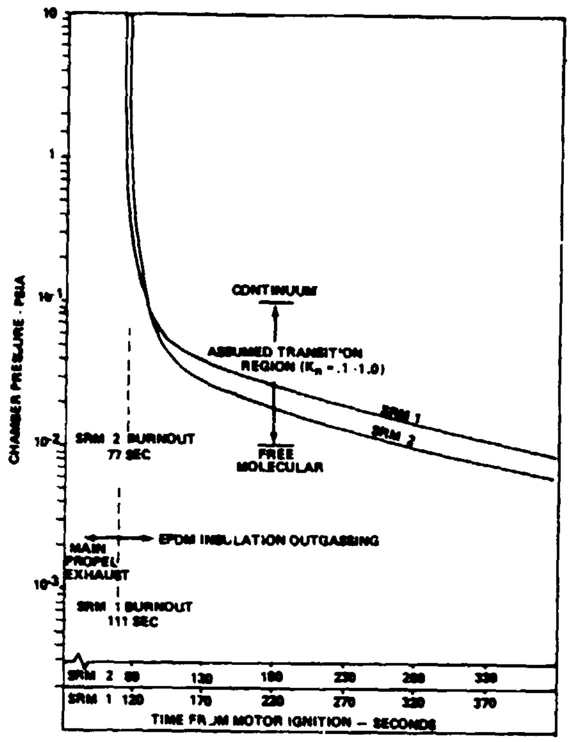

Figure 7: SRM1 & 2 Chamber Pressure Decay – Burnout Transient

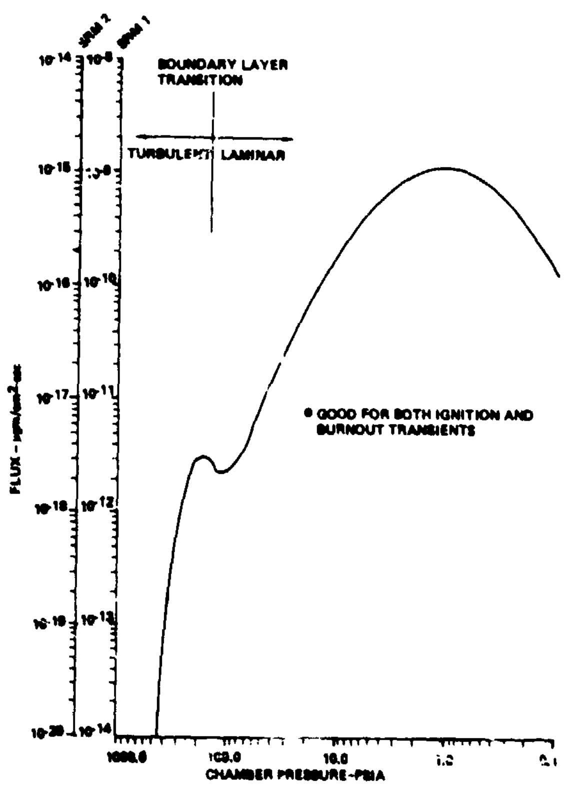

Figure 8: SRM Contamination Flux

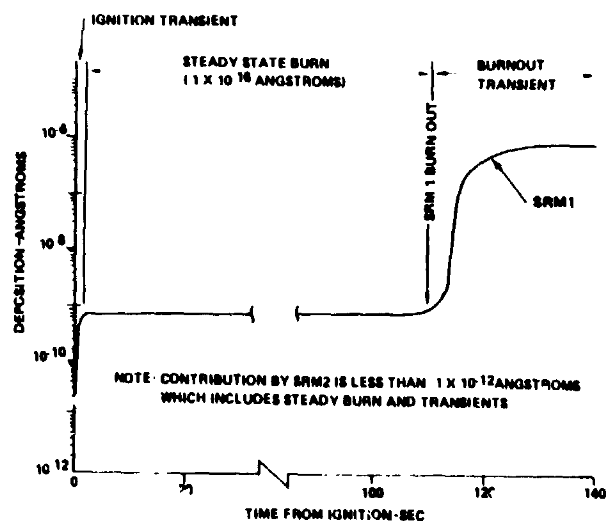

Figure 9: Contamination During Main Motor Burn

Table IV: Summary of Contamination to the Spacecraft During IUS Free Flight Deposition in Micrograms/cm^2

FLIGHT OPERATION / CONTAMINATION TRANSPORT	SRM1 BURN	SRM1 RETRO-GRADE	SRM1 HEAT SOAK	SRM2 BURN	SRM2 RETRO GRADE	SRM2 HEAT SOAK	S/C IUS SEP	TOTAL
SELF-SCATTERING	1×10^{-8}	0	0	1×10^{-12}	0	0	0	1×10^{-8}
AMBIENT SCATTERING	7.5×10^{-6}	(2) 1.5×10^{-2}						1.5×10^{-2}
DIRECT FLUX	0	0	6×10^{0}	0	0	(1) 4.2×10^{-1}	0	4.2×10^{-1}
TOTAL	7.51×10^{-6}	1.5×10^{-2}	6×10^{-6}	1×10^{-12}	0	4.2×10^{-1}	0	

(1) CONSERVATIVE ESTIMATE FROM KEVLAR-EPOXY MOTOR CASE WHEN HEATED TO 500°F 25 TO 30 MINUTES AFTER MOTOR BURNOUT.

(2) BASED ON 180 SEC. DELAY IN RETRO MANEUVER FOLLOWING SRM1 BURNOUT. NO DELAY WOULD PRODUCE 2.6 MICROGRAMS/CM^2.

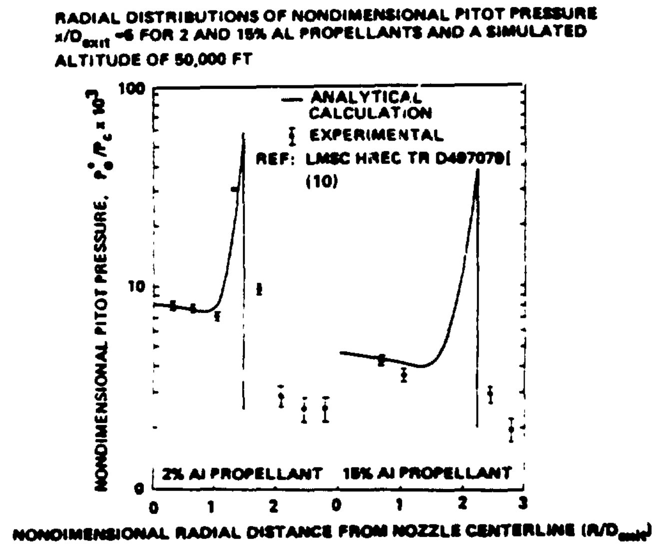

Figure 10: Pip Computer Code Verification – Pitot Pressures

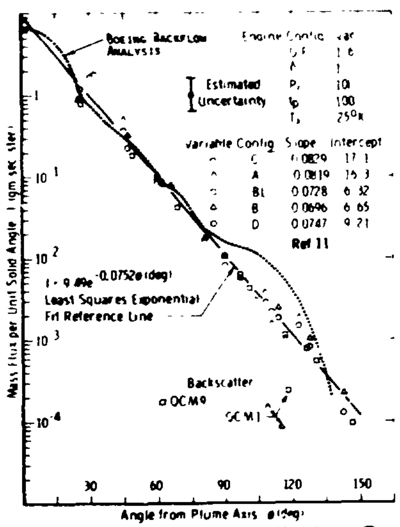

Figure 11: Boeing Backflow Prediction Correlation with Experimental Data (AEDC)

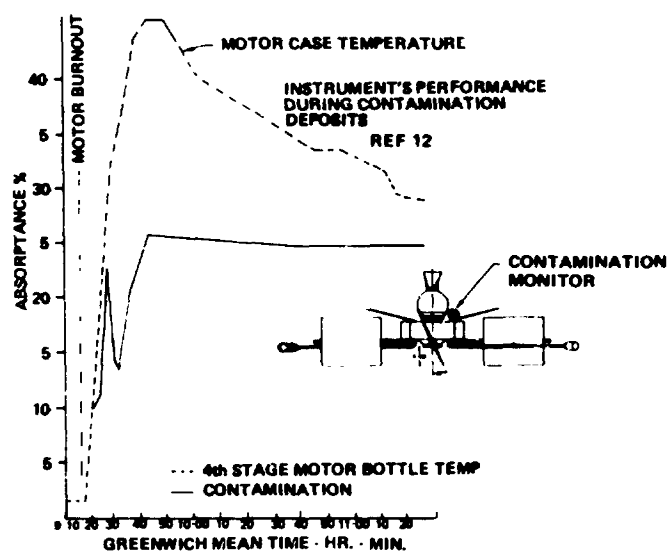

Figure 12: AIMP-E Satellite Flight Data